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The Design Of A High-Q, MACH-5 Nozzle For The NASA Langley 8-Foot HTT

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A new nozzle has been designed for the NASA Langley Research Center 8-Foot High Temperature Tunnel. The new nozzle was designed with a Mach-5 exit flow at a Mach-5 flight-enthalpy test condition and has a smaller throat area than the existing Mach-5 nozzle which significantly increases the range of dynamic pressures that can be achieved in the facility. The nozzle was designed using the NASA Langley IMOCND computer program which solves the potential equation using the classical method of characteristics. Several axisymmetric nozzle contours were generated and evaluated using viscous computational fluid dynamics. A number of items were considered in the evaluation, including flow uniformity, thermal and structural design, manufacturing schedule and cost. Once the final contour was selected, studies were done to determine the effects of manufacturing irregularities (steps and cavities at joints). These studies were done to develop manufacturing specifications and assembly tolerances.

I. Introduction

The NASA Langley 8-Foot High Temperature Tunnel[1] (8-Ft. HTT) is a combustion-heated hypersonic blowdown-to-atmosphere wind tunnel that provides flight enthalpy simulation for Mach numbers of 4, 5 and 7 through an altitude range from 50,000 to 120,000 feet. The open-jet test section is 8-ft. in diameter and 12-ft. long. The test section will accommodate large air-breathing hypersonic propulsion systems as well as structural and thermal protection system components. Stable wind tunnel test conditions can be provided for a duration of up to 120 seconds. The test medium is the combustion products of air and methane that are burned in a pressurized combustion chamber. Oxygen is added to the test medium for air-breathing propulsion tests so that the test gas contains twenty one percent molar oxygen. The facility currently has three nozzles with exit Mach numbers of 4, 5 and 7. For the Mach 4 and 5 test conditions a mixer section is added to the facility flow path downstream of the burner where ambient-temperature air is added and mixed with the hot gas from the burner to achieve the desired test enthalpy (see Figure 1).

The 8-Ft. HTT has recently been used to conduct a series of tests of hypersonic engine systems. Recent tests have included the NASA X-43A engine (Hyper-X), the Office of Naval Research HyFly Dual Combustor Ramjet Engine, and most recently the joint USAF/NASA testing of the X-43C Ground Demonstrator Engine-2. All of these tests have utilized facility components that were designed and fabricated in the mid 1980's.

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Current hypersonic engines are being designed to operate much lower in the atmosphere than the engine systems that the facility was originally designed to test. There are existing test requirements for the upcoming Air Force Research Laboratory (AFRL) Scramjet Engine Demonstrator (SED) program at Mach-5 test enthalpy at higher dynamic pressure than the existing Mach 5 nozzle can provide. The maximum achievable dynamic pressure with the existing nozzle, (96 inch nozzle exit diameter) is about 1150 pounds per square ft (psf). However, this dynamic pressure can be increased by building a new nozzle that is scaled down in size. The scaled down nozzle has a significantly smaller throat and exit diameter, but the smaller throat allows higher upstream stagnation pressures to be reached for the same test gas flow rate.

At the request of the SED program manager a parametric study was undertaken which showed that reducing the Mach-5 nozzle throat diameter from 16.9 inches to 9.2 increases the nozzle exit dynamic pressure from 1150 psf to 3300 psf at demonstrated facility test flow rates and at the maximum pressure rating of the ASME code stamped facility mixer. With a nozzle of this size the facility would be able to achieve the Mach-5 flight-enthalpy, 2500 psf test condition required by the SED program. In order to accommodate higher dynamic pressure tests in the 8-Ft HTT, a decision was made to design and build a new Mach-5 nozzle. As part of the new design effort, a number of different aspects were considered including: flow uniformity, thermal and structural design, level of difficulty to manufacture, schedule and cost. The role of each of these items is discussed in the following sections as well as design decisions which were influenced by them.

II. The Design Method - MOC

The new nozzle is required to accelerate the test gas from the nearly stagnate conditions in the facility mixer to a Mach number of 5. The exit flow should not only be at the correct Mach number, but also have a minimum of non-uniformity. It is assumed that the test gas in the plenum is uniform in all properties. Based on these requirements and assumptions, a nozzle design code which solves the compressible potential equation using the method of characteristics (MOC) was chosen as the design tool. The NASA Langley code, named IMOCND (Irrotational MOC for Nozzle Design), has been used to design both two-dimensional and axisymmetric nozzles for use in the NASA Langley Research Center Direct Connect Supersonic Combustion Test Facility.

The method of characteristics is very useful in supersonic nozzle design as the requirement for uniform exit flow provides a boundary condition which determines the shape of the re-compression portion of the contour. This is accomplished by tracing information from the nozzle exit, upstream along characteristic lines, to the nozzle surface and setting the surface slope to cancel waves which intersect the surface from farther upstream. With the IMOCND code, the user provides the target Mach number and the functional form of the initial expansion surface (e.g. Gaussian curve, circular arc) and the code generates a nozzle contour which perfectly cancels all of the expansion waves. Below is a brief explanation of the method of characteristics and its implementation in IMOCND.

For inviscid, irrotational flow the governing equations can be reduced to the compressible potential equation. This is a single, 2^{nd} order partial differential equation. In two-dimensional/axisymmetric supersonic flows there exist two curves (called characteristic lines) along which the partial differential equation reduces to an ordinary differential equation (a slightly different equation for each line). The ordinary differential equations are called compatibility equations. For calorically perfect gas flows the compatibility equations are:

$$d\left(\nu \pm \theta\right) - \delta\left[\frac{\sin\mu\sin\theta dr}{r\sin\left(\theta \mp \mu\right)}\right] = 0 \qquad (1)$$

where θ is the flow angle, ν is the Prandtl-Meyer function, μ is the Mach angle and r is the radial distance from the centerline. The term δ in equation 1 is zero for two-dimensional flow and one for axisymmetric flow. For the special case of two-dimensional flow the second term drops out and the equation can be integrated exactly. The compatibility equations hold along left and right running characteristic lines (also called Mach

lines) given by

$$\frac{dr}{dx} = \tan\left(\theta \pm \mu\right) \tag{2}$$

where x is the axial coordinate. These equations are solved numerically by discretization of the equations on a grid created by intersecting characteristic lines, such as that shown in Figure 2. The discretization of the equations is that given by Anderson[2] and involves two adjacent points in the grid. (Each equation involves its own pair of grid points.) Differentials in the equation are replaced with first order backward differences and all other terms are averages between the two points. The discretization of equation 1 is

$$\left[(\nu \pm \theta)_2 - (\nu \pm \theta)_1\right] - \delta\left[\frac{\sin\overline{\mu}\ \sin\overline{\theta}\ \ (r_2 - r_1)}{\overline{r}\ \sin\left(\overline{\theta} \mp \overline{\mu}\right)}\right] = 0 \tag{3}$$

where the over-bar denotes the arithmetic average:

$$\overline{\phi} = \frac{\phi_1 + \phi_2}{2} \tag{4}$$

Each equation is discretized along the appropriate characteristic curve and involves one known point and one unknown point, which results in a system of coupled non-linear algebraic equations that are solved simultaneously using a Newton-Raphson method. Because the flow is supersonic, solution points can be computed sequentially. For nozzle design, the solution at points that depend on the inflow and the user-specified expansion portion of the nozzle (the blue region in Figure 2, also called the kernel) are computed first. The turning contour and the points dependent on it (the green region in Figure 2) are then computed. The solution procedure at each point depends on the type of point being solved; interior, centerline, surface point, etc. Along the centerline, one of the compatibility equations is replaced with the boundary condition $\theta = 0$. Similarly, at the surface of the expansion section one of the compatibility equations is replaced with the boundary condition $\theta =$ surface slope. The surface in the re-compression section is determined by forcing conservation of mass at each axial location and the requirement of uniform exit flow.

III. Initial Aerodynamic Design

The design conditions for the new nozzle were: an exit Mach number of 5, a total pressure of 840 psi and a total temperature of 2140 Rankine. This corresponds to a Mach-5 flight enthalpy. The chemical composition of the test gas is given in Table 1. The only design constraint was that the nozzle fit between the mixer and

Specie	Mole Fraction
N_2	0.644
O_2	0.216
H_2O	0.088
CO_2	0.044
Ar	0.008

Table 1. Test gas species composition.

the test section, a distance of 434 inches. Two nozzle contours were proposed, one long and the other short with a cylindrical constant-area extension. Both nozzles were designed by iterating the IMOCND code with a viscous CFD calculation[3] (using the VULCAN[4] CFD code) in order to include the effect of the boundary layer displacement and to achieve the target exit Mach number. (Since these were preliminary designs, only 2 iterations were done.) The long nozzle had a very large throat radius of curvature (920 inches), a very gradual variation in the contour shape, a length of 387 inches from the throat to the exit and an exit diameter

of 52.8 inches. The CFD solution computed with the VULCAN CFD code using a grid of 501 x 97 and a $k-\omega$ turbulence model with wall functions and a surface temperature of 800 F, indicated an exit Mach number of 4.95, a very uniform exit flow with an exit pressure variation of only 0.05 percent difference and a core size of 41 inches in diameter. For this paper the percent difference is defined as

$$\text{Percent difference} = 100\left[\frac{P_{max}-P_{min}}{\frac{1}{2}\left(P_{min}+P_{max}\right)}\right]$$

Figures 3 and 4 show contours of the Mach number and of the $\log_{10}$ of the static pressure in the nozzle. (Log_{10} is used for the pressure because it better distributes the contours in the nozzle.) Figure 5 shows the Mach number and static pressure profiles at the nozzle exit. At the exit, the boundary layer is almost 6 inches thick.

The short nozzle had a throat radius of curvature of 23 inches, a length of 206 inches and an exit diameter of 54.5 inches. Figures 6, 7 and 8 show the Mach number contours, $\log_{10}$ of the pressure and the exit profiles computed using CFD on a 501 x 97 grid. Although the nozzle is shorter, it still has very uniform exit flow with only a 0.99 percent difference in the static pressure. The core flow size is 45 inches and the exit Mach number is 5.01. The difference in the levels of non-uniformity between the two contours is due to the more gentle, quasi-1D expansion of the longer nozzle. After presenting these two options to the design team and the AFRL SED program manager, the short nozzle was selected based on the assumption that it would be easier and faster to manufacture than the long nozzle and its fabrication cost would be less.

IV. Final Aerodynamic Design

Although the flow quality for the short nozzle was deemed acceptable, some additional work was done to optimize the contour. One of the issues not taken into account in the initial design was the effect of the variation of the ratio of specific heats, γ, which can be significant in high enthalpy facilities such as the 8-Ft HTT. For the current design condition, γ varies between 1.30 in the mixer and 1.39 at the nozzle exit. Since the IMOCND code uses only a single value of γ, a search was made for the best overall value. This was accomplished by generating four nozzle contours, each with a different value of γ, and using axisymmetric viscous CFD calculations with a variable γ thermodynamics model (VULCAN) to determine the value of γ which produced the least exit flow non-uniformity. For each contour the percent difference in the static pressure at the exit was computed and the resulting values fit with a polynomial. The value of γ corresponding to a minimum percent difference was then interpolated from the curve. The four values of γ used in this exercise were 1.33, 1.34, 1.35 and 1.36. Each of the contours was generated with an initial expansion section defined by a Guassian curve with a throat radius of curvature of 23 inches. None of the contours was smoothed in a post processing step. In order to reduce computational time only the nozzle, without the cylindrical extension, was included in the computational domain. For each case a grid of 385 x 97 was used and the L2 norm of the solution error reduced by a minimum of 9 orders of magnitude. The $k-\omega$ model was used with wall functions and a surface temperature of 800 F. The results are shown in Table 2. The polynomial fit of the four cases had a minimum value of percent difference at a γ of about

Ratio of Specific Heats	% Difference
1.330	6.05
1.340	3.68
1.350	0.96
1.353	0.61
1.360	2.12

Table 2. Exit static pressure variations for nozzles designed with values of γ between 1.33 and 1.36

1.353. This case was also run using CFD with the result that the nozzle had a 0.61 percent difference in static pressure at the nozzle exit.

As a last optimization step, the final contour was smoothed to reduce small perturbations in the contour shape introduced by numerical errors in the MOC solution procedure.[5] The CFD solution for the final nozzle contour and cylindrical extension was converged more than 9 orders of magnitude on a 1025 x 193 grid using the $k-\omega$ turbulence model solved to the wall with a surface temperature of 800 F. Contours of Mach number and the $\log_{10}$ of the static pressure are show in Figures 9 and 10 and the exit profiles are shown in Figure 11. The nominal exit Mach number is 4.99, with a variation of only 0.088 percent difference. The nominal exit static pressure is 1.334 psi with a variation of 0.46 percent difference. Because the new nozzle is able to produce a test flow with a higher dynamic pressure (Q), the new nozzle has been designated the M5HiQ nozzle.

The final nozzle contour was also evaluated at two other conditions. The first condition is at the same total temperature as the design condition (2140 R) but at a lower pressure of 634 psi. For this case the gas composition remained the same as the design condition. Figure 12 shows the Mach number and static pressure profiles at the nozzle exit. As in the design case, the flow is very uniform with a variation of only 0.47 percent difference in static pressure. The second condition corresponds to a Mach 5.5 flight enthalpy. The total temperature is 2423 R and the total pressure is 668 psi. (At this total temperature, the pressure is limited by the mass flow limit of the facility.) The gas composition for this condition is given in Table Figure 13 shows the Mach number and static pressure profiles at the nozzle exit. For this condition the nozzle has an exit Mach number of 4.92 and a non-uniformity in static pressure of 2.24 percent difference. 3.

Specie	Mole Fraction
N_2	0.644
O_2	0.216
H_2O	0.088
CO_2	0.044
Ar	0.008

Table 3. Test gas species composition for a Mach 5.5 enthalpy test condition.

V. Thermal/Structural Design

Once the design of the nozzle contour was finalized, another CFD calculation was done on the contour with an isothermal surface temperature of 600 Fahrenheit. The surface heat fluxes with wall temperatures of 600 and 800 degrees and the surface pressures for the two cases were passed to the thermal/structural design team for the thermal/mechanical design. The two heat fluxes at the two different wall temperatures were used to create a linear fit at each axial point of heat flux as a function of wall temperature. This fit was used in an unsteady thermal analysis which allowed the heat flux to vary as the surface temperature changed in time. Figure 14 shows the axial heat flux distribution for the constant wall temperature cases. The thermal analysis showed that the nozzle could run for 90 seconds at a Mach-5 flight-enthalpy condition without cooling if the throat region was made of five inch thick Inconel. The analysis also indicated that small cracks will develop during cool down, but they will be pressed together during heat up.

Due to its length, the heat load in the throat region, the need to install and integrate the nozzle with existing facility hardware and manufacturing considerations, the nozzle was designed as eight major components. These are the throat insert; extensions 1, 2A, 2B, 3, 4, 5 and a connecting plate (see Figure 15). Fortunately the throat component is small enough to fit inside of the pressure shell of the larger Mach-5 nozzle which means that the new throat component does not have to be code stamped as a pressure vessel, reducing production cost. Similarly, extensions 3, 4 and 5 sit inside of the downstream pressure shell of the

existing nozzle and do not have to be code stamped as pressure vessels. The connecting plate was designed to adapt the new nozzle hardware to the existing tunnel pressure shell, to allow for insertion and removal of extensions 3, 4 and 5, and to provide radial support for the nozzle components. Installation constraints required that extensions 1, 2A, 2B and the connecting plate form a pressure boundary for the nozzle flow. This required that these components be designed and fabricated according to the ASME Boiler and Pressure Vessel code. The material for all extensions and the connecting plate was designated 304 stainless steel.

The initial throat design studies were based on the assumption that Inconel 725 or 618 alloy be used to cast the uncooled throat section. Unfortunately neither Inconel alloy was available in sufficient quantities for this project, nor were casting providers available to produce the large refractory metal castings. To counter these issues, the throat section was redesigned as a brazed, three section assembly made from Incoloy 800HT material. Unlike previous lower dynamic pressure nozzles, high heat fluxes extend well past the throat section and through nearly the entire length of extension 1. This required a thick-walled heat sink design, similar to the throat component, to be carried through most of extension 1, resulting in a 7 inch thick weld at the upstream flange. All welding was accomplished using a submerged arc process, with vibration stress relief prior to final machining.

Typically, adjacent segments of high performance nozzles are machined simultaneously to prevent steps or discontinuities at the joints. Analysis showed that the overall schedule could be dramatically improved if the individual segments could be machined separately. However this introduced the possibility of joints which do not fit perfectly together, resulting in steps at the interface. A forward facing step is less desirable than a rearward facing step as it creates a normal shockwave followed by an expansion fan, while a rearward facing step creates an expansion fan followed by an oblique shockwave (less overall loss). To prevent forward facing steps, and to control the magnitude of aft-facing steps, a concept was investigated to intentionally design rearward facing steps into the downstream component of each joint. However, this introduced questions about the aerodynamic effects of steps and gaps at the joints between segments. Early in the structural design cycle a nozzle contour with two joints was considered. (The constant area section would add additional joints.) The first joint was 27.6 inches downstream of the throat and the second joint was 181.6 inches downstream of the throat. In order to determine the aerodynamic effect of joints and to determine acceptable step tolerances, four CFD calculations were done with rearward facing steps of two different heights at the two joint locations. The first pair of steps had heights of 0.025 and 0.050 inches and the second pair of steps had step heights of 0.050 and 0.100 inches. The steps were created by radially translating the contour downstream of the joint by the step height. At the 27.6 inch joint a step height of 0.025 inch produces an exit pressure variation of 3.24 percent difference and a step height of 0.050 inch produces an exit pressure variation of 5.72 percent difference. The Mach number and pressure profiles at this location are shown in Figure 16 and the results are summarized in Table 4. Recall that a rearward facing step creates an expansion fan followed by an oblique recompression shockwave. Both of these reflect off of the opposite surface and are dissipated only slightly along the length of the nozzle (see Figure 17). Figure 18 shows the exit profiles for the two step

Location downstream of throat (in inches)	Step height (in inches)	% Difference in P at exit
27.6	0.025	3.24
27.6	0.050	5.72
181.6	0.050	2.05
181.6	0.100	4.00

Table 4. Pressure variation introduced by steps at 27.6 and 181.6 inches downstream of the throat.

heights at 181.6 inches downstream of the throat. Although the steps at this location are larger than those at 27.6 inches, the pressure variation is less due to the thicker boundary layer. For the 0.050 inch step the pressure variation was 2.05 percent difference and for the 0.100 inch step it was 4.00 percent difference. The

Mach number contours of Figure 19 shows a wave being generated from the 0.050 inch step at the 181.6 inch location, crossing the flowfield and exiting the nozzle before reflecting off of the opposite surface.

One other surface anomaly that was considered was a small cavity at the 181.6 inch joint created by a gasket between the two nozzle segments. For this case CFD was run with a 1/8 inch deep and 1/8 inch long cavity starting at the 181.5 inch location. Contours of the Mach number, shown in Figure 20, show that the fluid flows smoothly across the gap and that there is not any significant disturbance created by the cavity. This is because the boundary layer is several inches thick at this location and the cavity is relatively small in comparison.

Due to manufacturing and installation constraints, the total length of the final design was broken up into 8 segments with 7 joints. Based on the CFD results, the step height allowances were graduated, with smaller allowances near the throat and larger steps allowed as the boundary layer thickness developed along the length. These values are listed in Table 5.

Location downstream of throat (in inches)	Step height allowance (in inches)
27.6	0.010
87.6	0.015
132.6	0.020
176.9	0.025
176.9+	0.025

Table 5. Rearward facing step height allowances for the joints between segments.

VI. Summary

A new Mach-5 nozzle has been designed for the NASA Langley 8-Ft HTT for operation at a Mach-5 flight-enthalpy test condition. The new nozzle has a smaller throat than the existing Mach-5 nozzle which will enable testing at a dynamic pressure of up to 3300 psf. The new nozzle was designed with the Langley IMOCND code which solves the compressible potential equation using the irrotational method of characteristics. A study was conducted to determine the optimum value of the ratio of specific heats to use in the nozzle design process. The final contour had a throat diameter of 9.2 inches, a length of 206 inches from the throat to the exit and an exit diameter of 54.5 inches. Since the new nozzle is shorter than the existing Mach-5 nozzle, a constant area duct will connect the nozzle exit to the test section. At the Mach-5 flight-enthalpy test condition the core flow at the exit of the constant area section is 41 inches in diameter. At this location the flow is very uniform with a static pressure variation of only 0.46 percent difference. At a Mach-5.5 flight-enthalpy condition the variation in the exit static pressure increases to 2.24 percent difference.

The 5-inch thick, heat-sink throat piece is constructed of Incoloy 800HT and fits inside of the pressure shell of the existing Mach-5 nozzle. The remainder of the nozzle is composed of six segments. The first three are ASME code stamped pressure vessels and the final three are inside the downstream pressure shell of the existing nozzle and are not code stamped. Joints between segments are designed with rearward facing steps with a graduated allowable step height based on distance from the throat. The nozzle is currently under construction and will be delivered, installed and calibrated in late summer or early fall of 2006.

References

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[2]Anderson, J. D., Jr., *Modern Compressible Flow With Historical Perspective*, McGraw-Hill Book Co., New York, 1982.

[3]Gaffney, R. L., Jr. and Korte, J. J., "Analysis And Design Of Rectangular-Cross Section Nozzles For Scramjet Engine Testing," AIAA paper 2004-1137, January, 2004.

[4]White, J. A. and Morrison, J. H., "A Pseudo-Temporal Multi-Grid Relaxation Scheme for Solving the Parabolized Navier-Stokes Equations," AIAA paper 99-3360, June, 1999.

[5]Gaffney, R. L., Jr., "Design Of A Mach-15 Total-Enthalpy Nozzle With Non-Uniform Inflow Using Rotational MOC," AIAA paper 2005-0691, January, 2005.

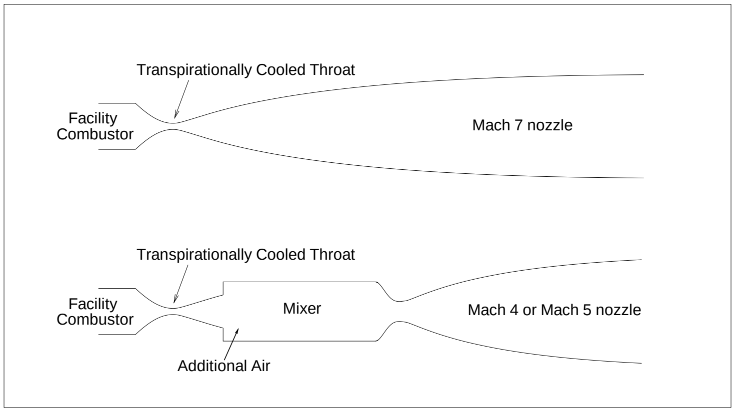

Figure 1. Facility nozzle configuration for Mach-7 operation (top) and Mach 4 and 5 operation (bottom)

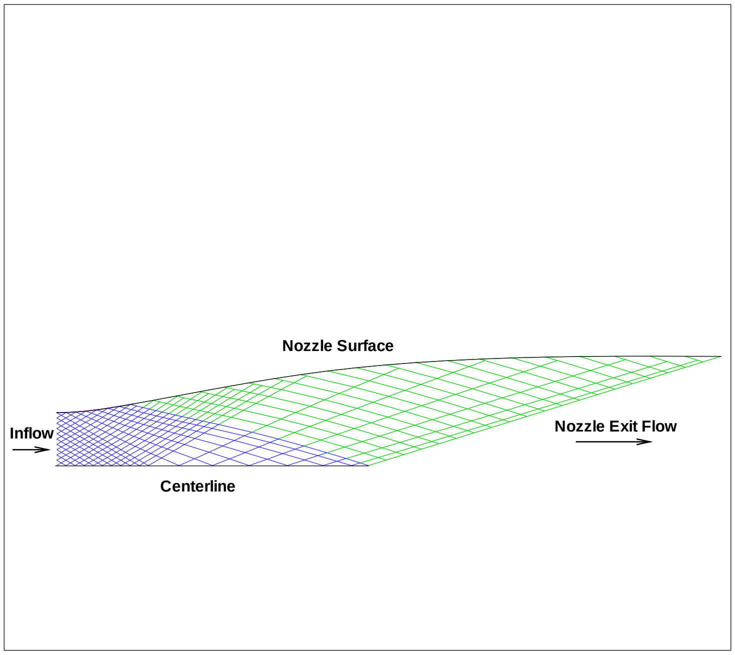

Figure 2. Example of a characteristic grid in a nozzle flow field.

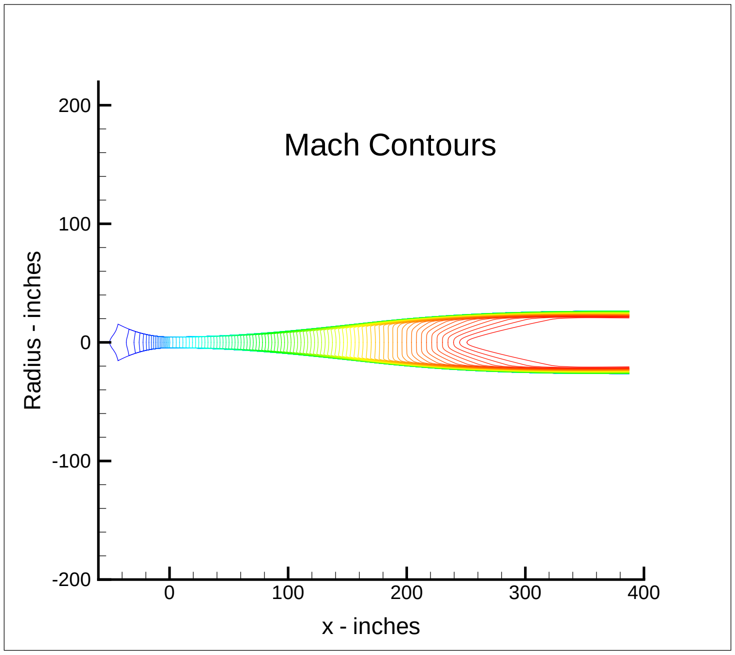

Figure 3. Mach number contours for the long nozzle.

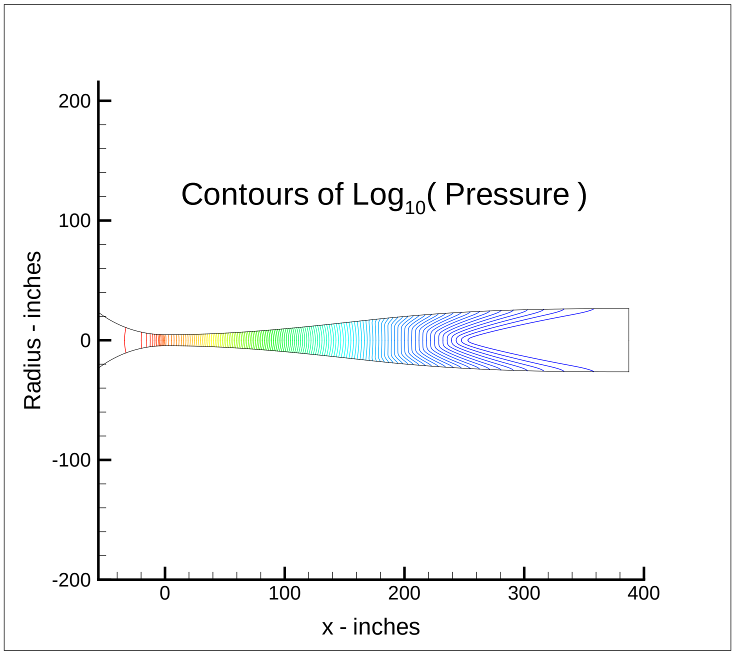

Figure 4. Static pressure contours for the long nozzle.

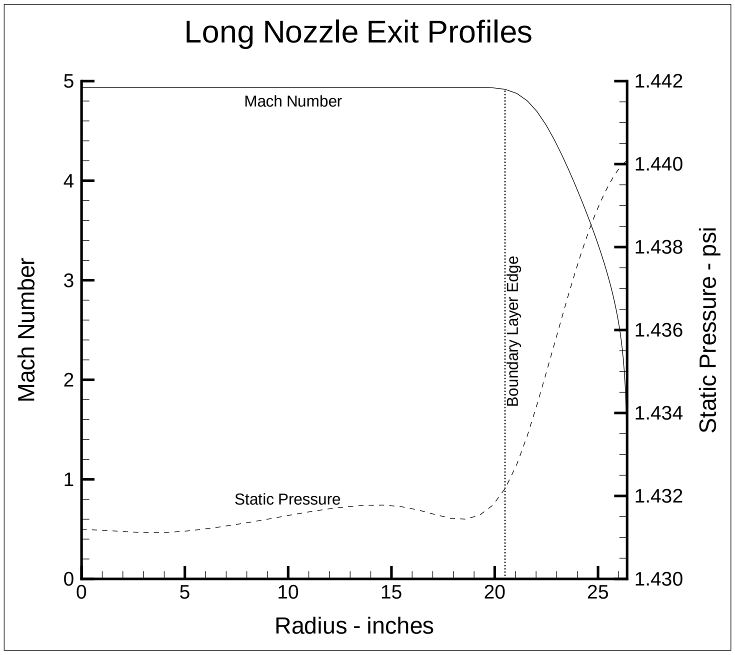

Figure 5. Mach number and static pressure profiles at the exit of the long nozzle.

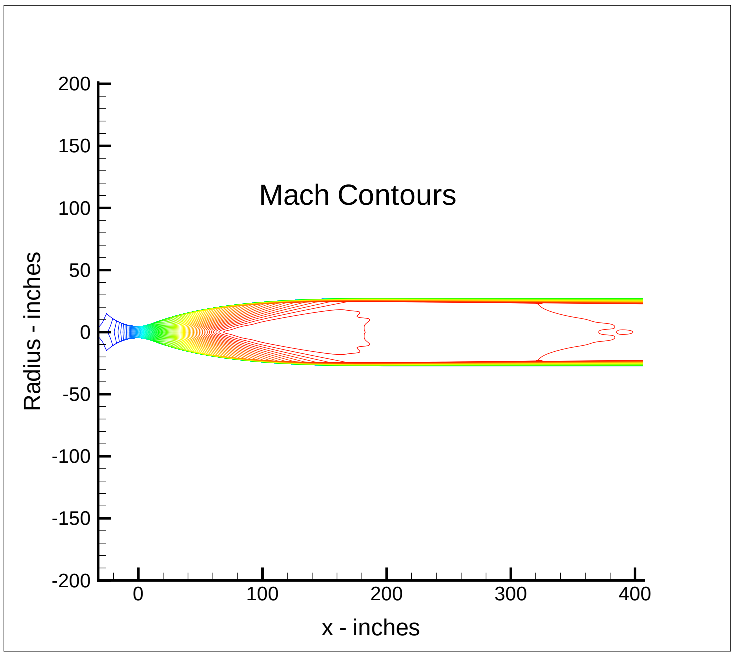

Figure 6. Mach number contours for the short nozzle.

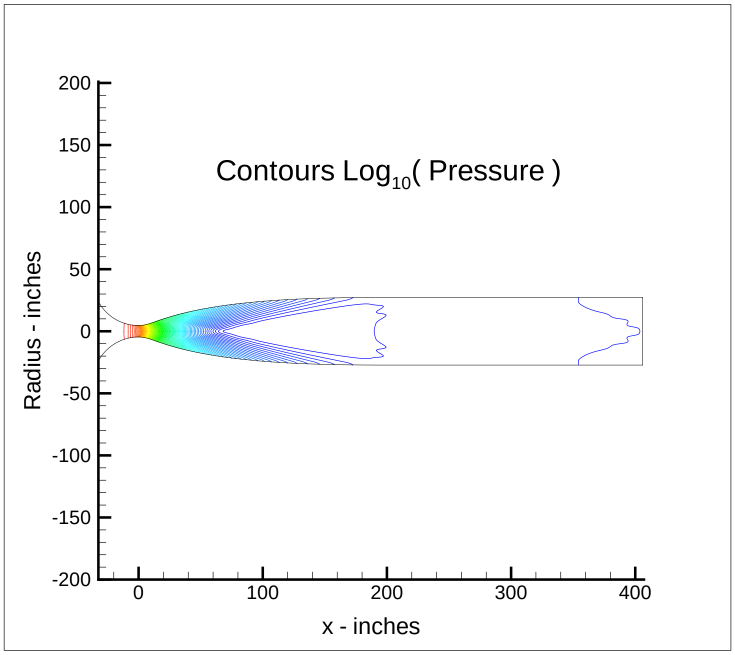

Figure 7. Static pressure contours for the short nozzle.

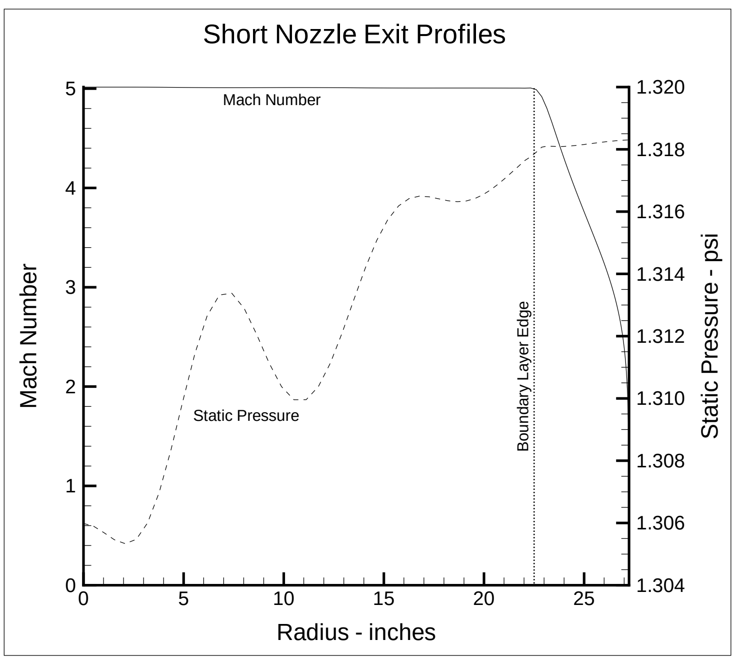

Figure 8. Mach number and static pressure profiles at the exit of the short nozzle.

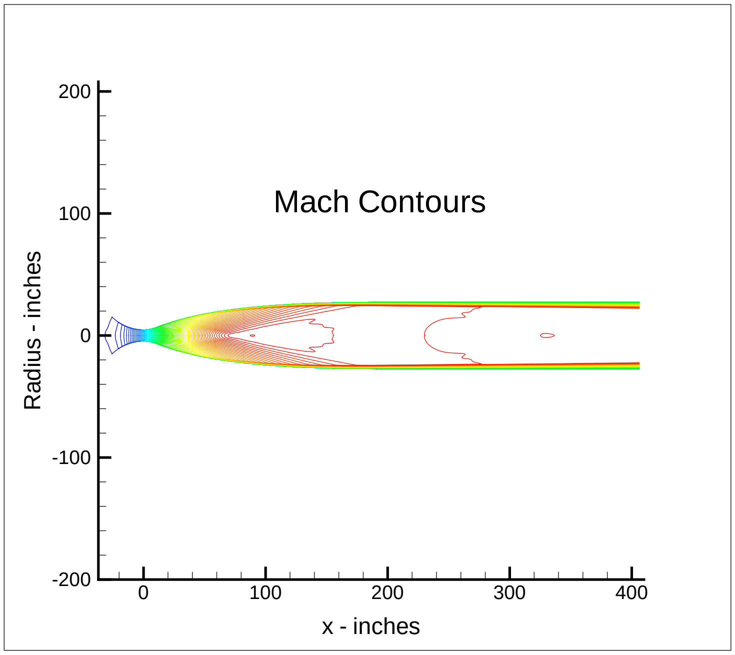

Figure 9. Mach number contours for the final nozzle.

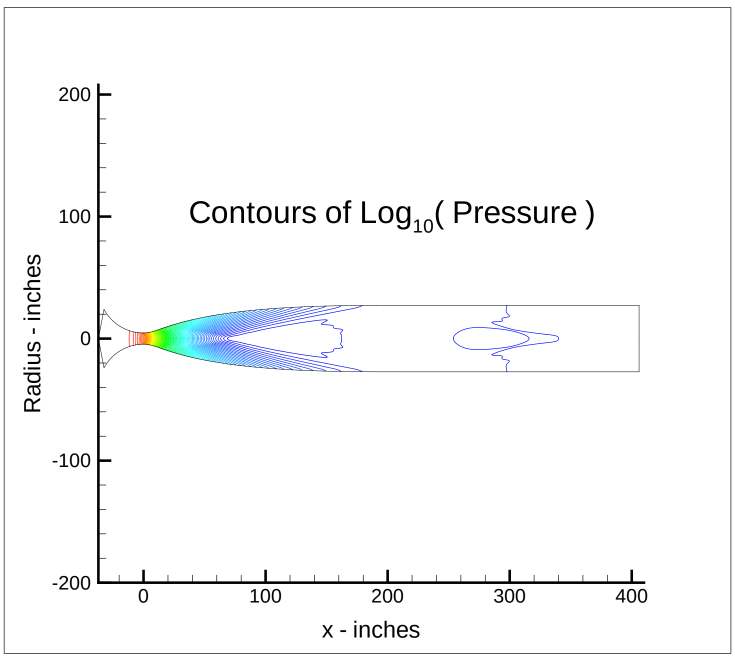

Figure 10. Static pressure contours for the final nozzle.

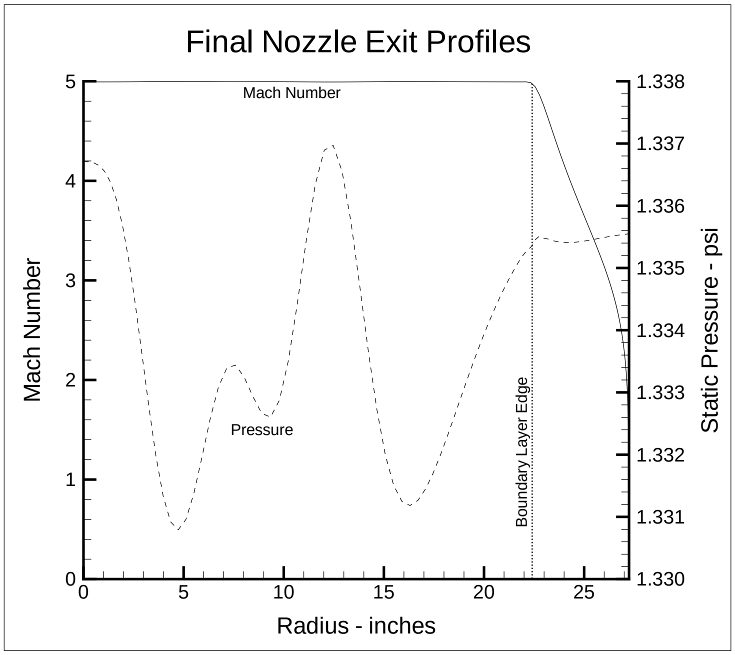

Figure 11. Mach number and static pressure profiles at the exit of the final nozzle.

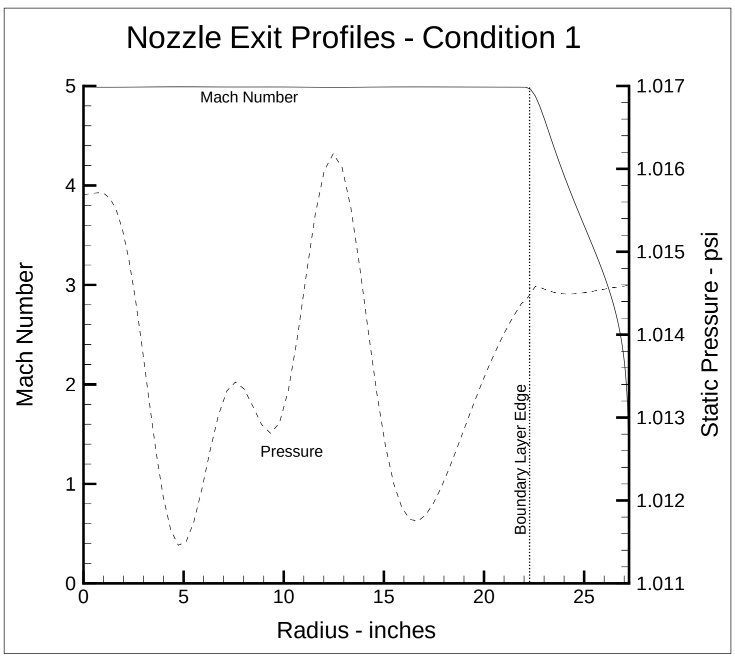

Figure 12. Mach number and static pressure profiles at the nozzle exit for test condition 1 (Mach-5 flight enthalpy).

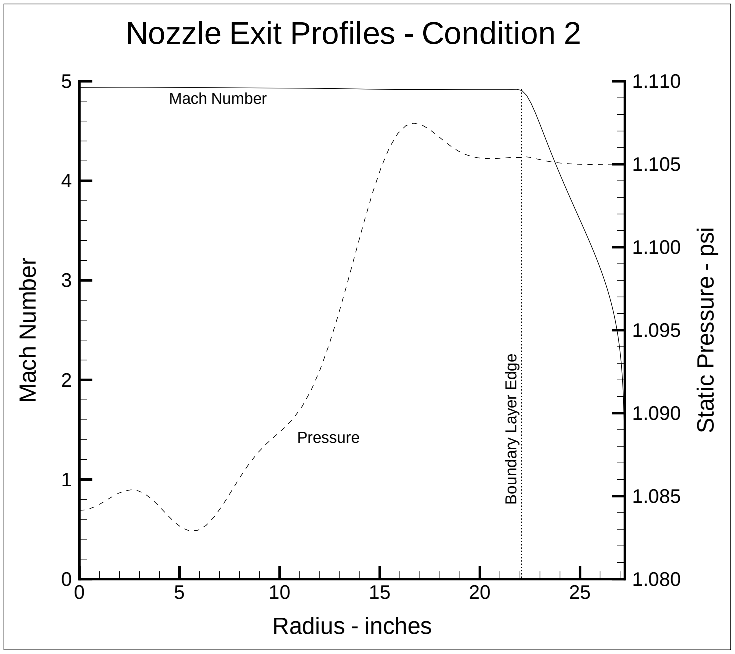

Figure 13. Mach number and static pressure profiles at the nozzle exit for test condition 2 (Mach-5.5 flight enthalpy).

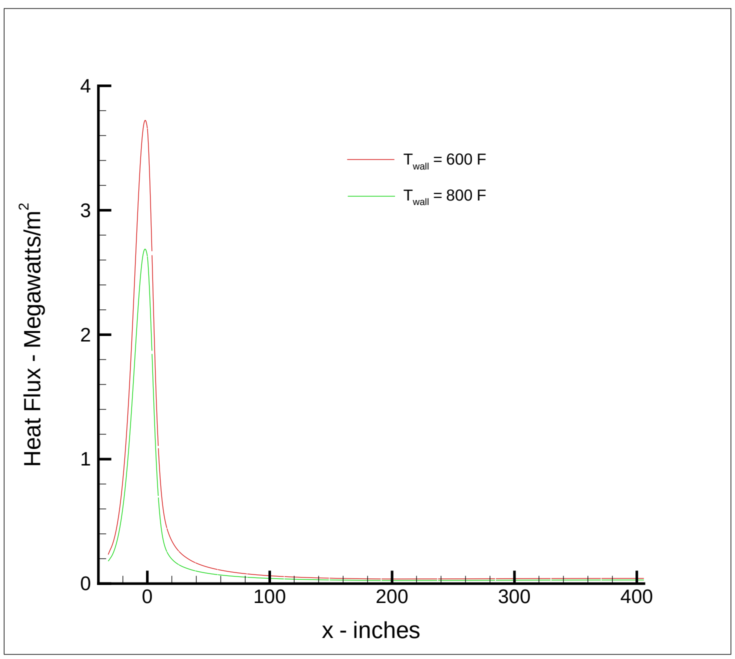

Figure 14. Axial heat flux distributions for wall temperatures of 600 and 800 F.

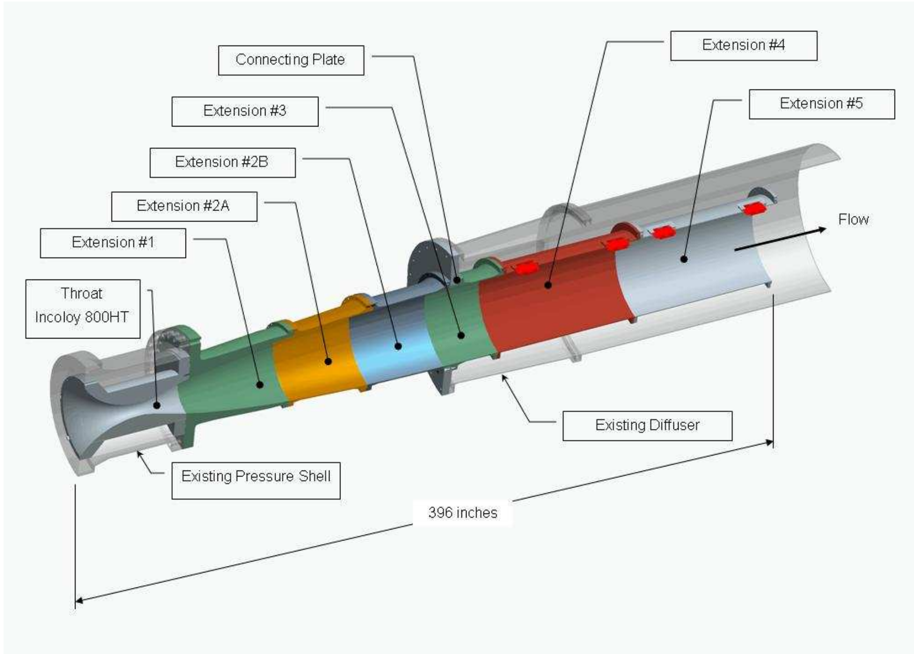

Figure 15. Components of the new Mach 5 nozzle.

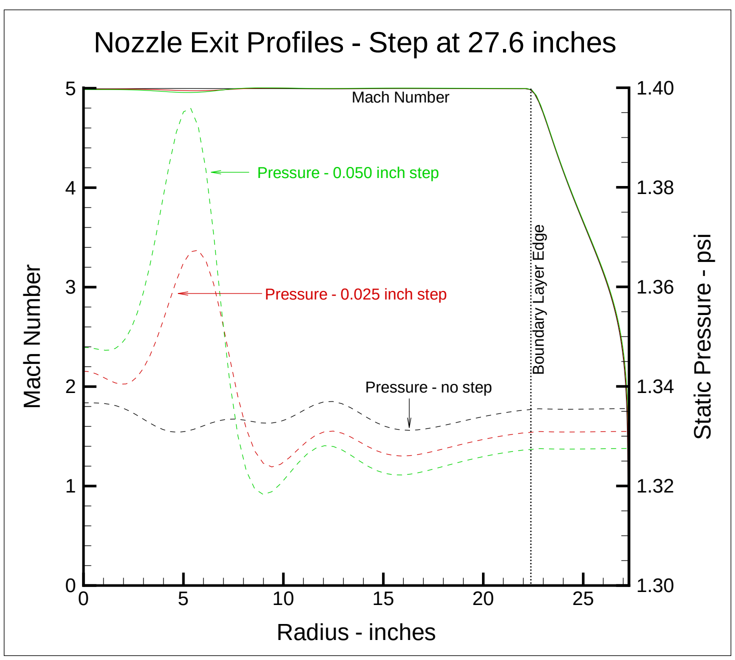

Figure 16. Mach number and static pressure profiles at the nozzle exit for steps 27.6 inches downstream of the throat.

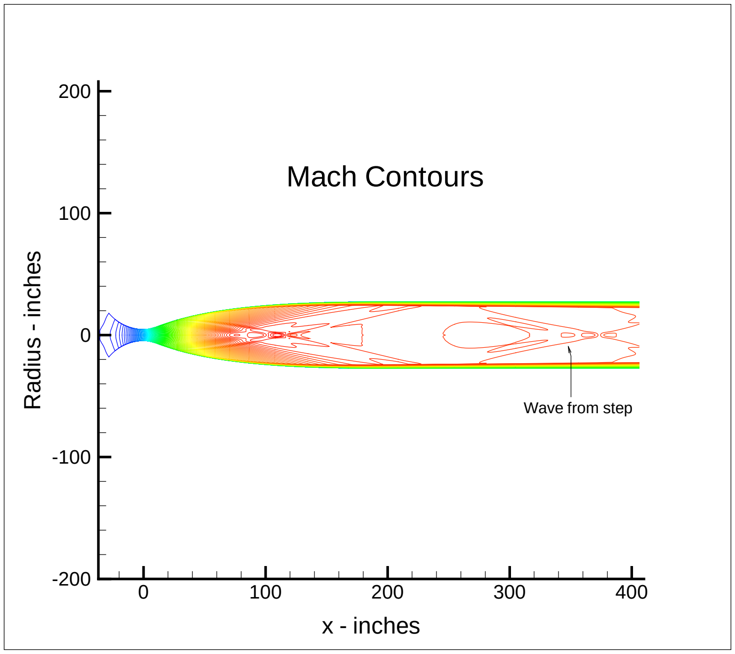

Figure 17. Mach number contours of the nozzle with a 0.025 inch step 27.6 inches downstream of the throat.

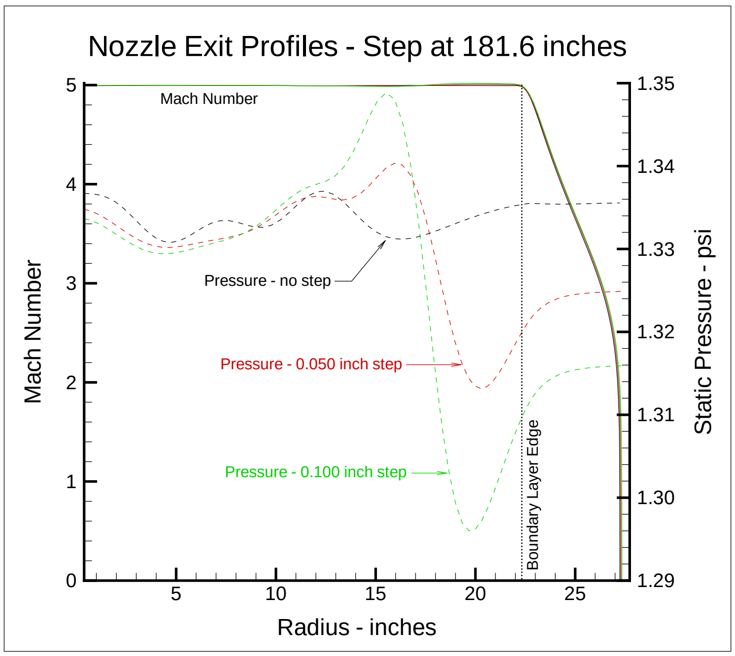

Figure 18. Mach number and static pressure profiles at the nozzle exit for steps 181.6 inches downstream of the throat.

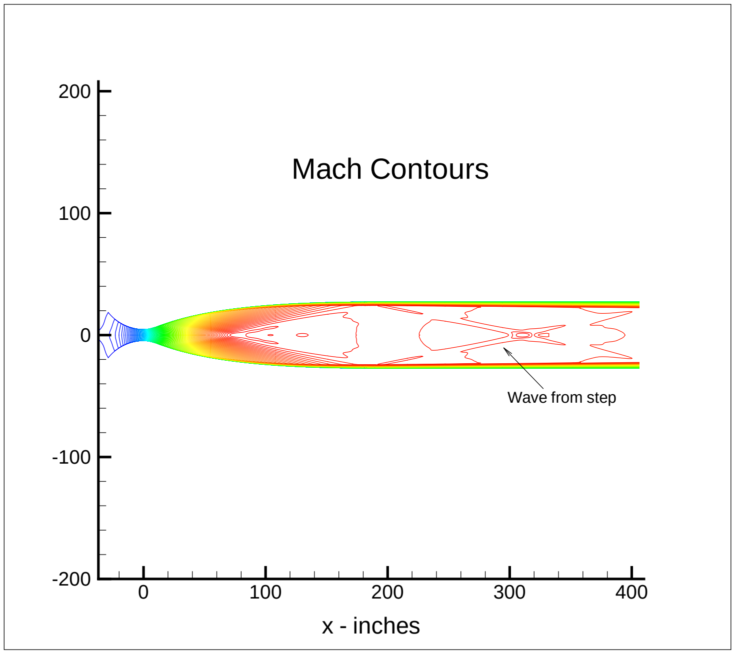

Figure 19. Mach number contours of the nozzle with a 0.050 inch step 181.6 inches downstream of the throat.

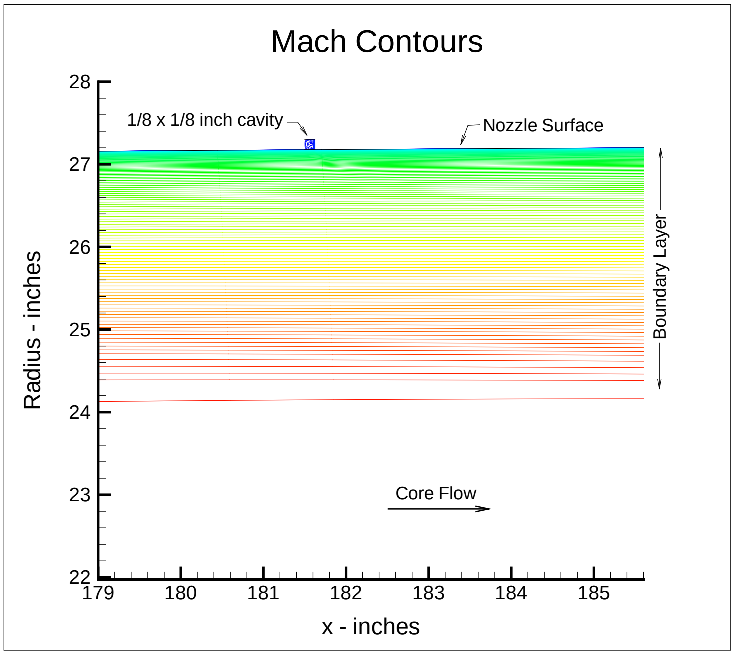

Figure 20. Mach number contours of the nozzle with a 1/8 x 1/8 cavity at 181.5 inches downstream of the throat.